Harvard-Smithsonian Center for Astrophysics

Precision Astronomy Group

MEMORANDUM

Date: 29 April 1998 TM98-06
To: Distribution
From: R.D. Reasenberg
Subject: Thermal analysis of a large Sun shield for FAME.

I. Introduction.

FAME is a full-sky astrometric survey instrument with a nominal mission accuracy of 50 μas for bright stars. The angle between the spacecraft spin axis and the Sun direction, ξ, is bounded at the high end by the Sun avoidance requirement and at the low end by the need for observational diversity, which reduces estimator degeneracy. For present purposes, the value will be assumed to be 45 deg. However, it must be remembered that this angle may vary over the mission.

The first objective of this study is to determine the effectiveness of a solar shield in reducing the thermal burden on the FAME instrument, and in particular, in reducing the time-dependent component of that thermal burden. Consideration must be given to a non-flat shield, to the variation of ξ (especially with the precession angle around the Sun), and to the Earth as a heat source. We find that, for a single-layer shield swept back enough to null the solar torque, the time-dependent part of the solar heating via the shield is an order smaller than the time-dependent part of the heating by Earth (for the spacecraft in its nominal 100,000 km orbit).

Section II contains the principal analysis of the memorandum. It addresses the heat that reaches the spacecraft from the Sun via reradiation by the flat shield. The analysis is extended to the case of a swept back shield in Section III, and the idea of an auxiliary shield is introduced in Section IV. At present, such an auxiliary shield is not seen to be needed. The thermal input from the Earth is addressed at a simplified level in Section V. Finally, some concluding remarks are offered in Section VI.

II. Analysis of the flat shield.

Figure 1 shows a cartoon of the spacecraft (instrument plus bus) with a round Sun-facing shield. Here we find the thermal input to a small patch on the cylindrical wall of the spacecraft from the warm back of the shield. As drawn, the spacecraft and shield have azimuthal symmetry around the spacecraft nominal spin axis (cylinder axis). We assume that the Sun light does not reach the cylinder directly (which is consistent with the dimensions given in Table 1). Then the temperature distribution is invariant under rotation around the spin axis provided only that the

Symbol	Nominal value	Description
r_1	1 m	radius of cylindrical spacecraft
r_2	3 m	radius of shield
ℓ_1	2 m	length of cylindrical spacecraft
R_1	0.8	reflectivity of Sun-facing spacecraft end (sun light)
R_2	0.9	reflectivity of shield (sun light)
ϵ_2	0.1	emissivity of shield (thermal IR)
S	1.36 kW/m^2	solar flux
σ	5.67×10^{-8} W m^{-2} K^{-4}	Stefan-Boltzmann constant

Table 1. Nominal quantities.

shield is flat ($\alpha = 0$, see next section) and the shield's Sun-facing surface has local optical properties that are azimuthally invariant. Under these conditions, the spacecraft, and particularly the instrument, cannot tell the Sun's azimuthal direction. For present purposes, we assume the nominal quantities shown in Table 1.

We start with a simple calculation to determine the thermal input at the cylinder wall, next to the shield. In this calculation, the heating of the shield by the thermal power coming from the cylinder is neglected. This is not a bad approximation since much of the spacecraft will likely be covered by multi-layer insulation (MLI) and therefore heat loss will be limited and radiant heat reaching the cylinder from the shield will mostly be reflected to space. Further, the back of the shield is also reflective, and little of the heat reaching it from the spacecraft would be absorbed. In the analysis that follows, I will ignore variations of surface optical properties with temperature and wavelength, except that I will distinguish (in principle) between properties for thermal IR (at ≈ 300 K) and for radiation from the Sun (at ≈ 5800 K).

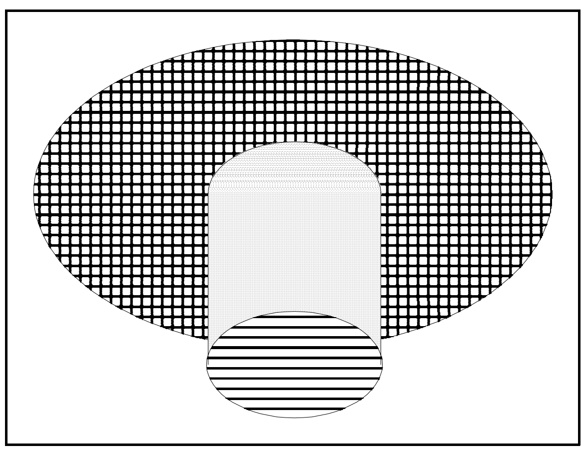
Figure 1. Spacecraft with shield.

The factors listed in Table 2 all reduce the thermal input to the cylinder wall from a nominal value of S, the solar flux. By taking their product, we obtain the net flux on the patch: 0.0175 S = 24 W/m^2. Without the shield, the flux would be S $\sin(\xi)$ Φ, where $\Phi = \{\cos(\nu)$,

Value	Description
0.7	cos(ξ), where ξ is the angle between the normal to the flat shield and the Sun direction
0.1	(1 - R_2), where R_2 is the reflectivity of the shield
0.5	shield radiates equally from both sides (shield assumed isothermal -- non-insulating -- and ϵ_2 assumed equal on both sides of shield)
0.5	shield fills half of the half space seen by the patch (This is the GCF discussed below.)

Table 2. Factors that decrease thermal input to the spacecraft.

0: depending on whether the patch is Sun lit or in shadow} and ν is the Sun's azimuth with respect to the patch. The average flux would be smaller by π than the peak value of S sin(ξ). Thus, the shield provides a 40 fold reduction from the maximum flux and, far more importantly, it provides a constant flux.

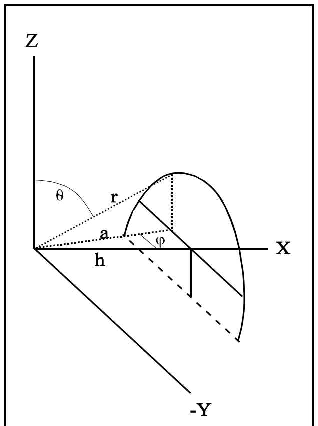

Figure 2. Geometry for calculation of GCF.

The corresponding calculation for a point on the cylinder wall at a distance *h* from the shield is more complicated. The geometry is shown in Fig. 2, where the patch of interest is at the origin of the coordinate system and is in the *X-Y* plane. The shield is at a distance *h* in the *X* direction, and the part of the shield visible to the patch has *Z* > 0.

Here it pays to introduce the standard approach of thermal analysts. The analysis assumes that the thermal radiation from all surfaces of interest is diffuse, i.e., the intensity (power per unit area per unit solid angle) is isotropic above the surface. The Geometric Configuration Factor[1] (GCF), $F_{1\text{-}2}$, is defined as the fraction of the energy leaving surface #1 that arrives at surface #2. It can be shown that, when surface #2 is small (strictly, a differential area), $F_{2\text{-}1}$ is the fraction of the weighted solid angle through which surface #2 receives energy from surface #1, where the weighting factor is cos(β) and β is the angle between the normal to surface #2 and the look direction. Further, $A_1F_{1\text{-}2} = A_2F_{2\text{-}1}$, where A_i is the area of surface i, i = {1,2}. Thus, an isothermal black surface of area A_d, insulated on the back, will be at equilibrium with incoming radiation at temperature, T_d, receiving and radiating power Q, for

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[1] See R. Siegel and J.R. Howell, *Thermal Radiation Heat Transfer*, Hemisphere Publishing, Washington, 1992, especially chapter 6.

$$Q = \sigma A_d T_d^4 = \sigma \sum_{i=1}^{N_s} F_{i-d} A_i T_i^4$$

$$T_d^4 = \sum_{i=1}^{N_s} F_{d-i} T_i^4 \qquad \textbf{(1)}$$

where N_s is number of surfaces that can be seen by the surface A_d.

For a patch (i.e., a differential area) on the cylinder surface shown in Fig. 1, $N_s = 2$. The surfaces are the back of the shield and free space.[2] To evaluate Eq 1, we could find the GCF for the shield (heating the patch), $F_{s\text{-}p}(h)$. However, it is more useful to evaluate $F_{p\text{-}s}(h) = I(h)$, where

$$I(h) = \frac{1}{\pi} \iint_{\text{shield}} \cos(\theta)\, d\Omega$$

$$d\Omega = \sin(\theta)\, d\varphi\, d\theta \qquad \textbf{(2)}$$

and the integral is over the visible part of the shield. By integrating over θ and making use of symmetry in φ, we obtain

$$I(h) = \frac{2}{\pi} \int_0^{\varphi_{max}} \left[\frac{\sin^2(\theta)}{2} \Bigg|_{\pi/2}^{\theta_0(\varphi)} \right] d\varphi \qquad \textbf{(3)}$$

where the required limits can be shown to be

$$\varphi_{max} = \arctan\left(\frac{\sqrt{r_2^2 - r_1^2}}{h} \right)$$

$$\theta_0(\varphi) = \arcsin\left(\frac{h}{\sqrt{(r_2^2 + r_1^2 + h^2)\cos^2(\varphi) - 2 r_1 \cos(\varphi) \sqrt{r_2^2 \cos^2(\varphi) - h^2 \sin^2(\varphi)}}} \right) \qquad \textbf{(4)}$$

[2] Free space is treated as a black surface. Here we take its temperature to be 0 K because the cosmic background radiation would not make a substantive contribution to the heat balance at ≈ 300 K.

The next integration was done numerically by using Mathcad.[3]® The result is shown in Fig. 3. We see that $F_{p-s}(1.5\,\mathrm{m}) \approx F_{p-s}(0)/3$ where h ≈ 1.5 m is of interest because it is about where the instrument aperture will be.[4] The radiation reaching the cylinder surface at h ≈ 1.5 m is then 8 W/m^2. If the surface were black and at 300 K, it would radiate 460 W/m^2. To radiate an additional 8 W/m^2, the temperature must increase by 1.3 K.

It is worth noting that the heat received at the instrument by reradiation from the shield depends on ξ. Consider the case of a spacecraft precessing under the torque due to solar radiation pressure. The value of ξ will change as the Sun moves toward and away from the spacecraft spin axis (which happens most when the spin axis is near the ecliptic). For a 10 day precession period, ξ will show a variation of ± 2.25 deg. (If small, the variation is [precession period]/[year × cosξ], converted to deg.) The corresponding fractional change in thermal power received at the spacecraft is ± 4%. Thus, the 8 W/m^2 received at h ≈ 1.5 m would have a variation of ± 0.3 W/m^2. Finally, since the variation of ξ and corresponding change in heat input are inversely proportional to the precession period, a slowly precessing spacecraft (e.g., with a 60 day precession period) would have a larger thermal variation at the precession period.

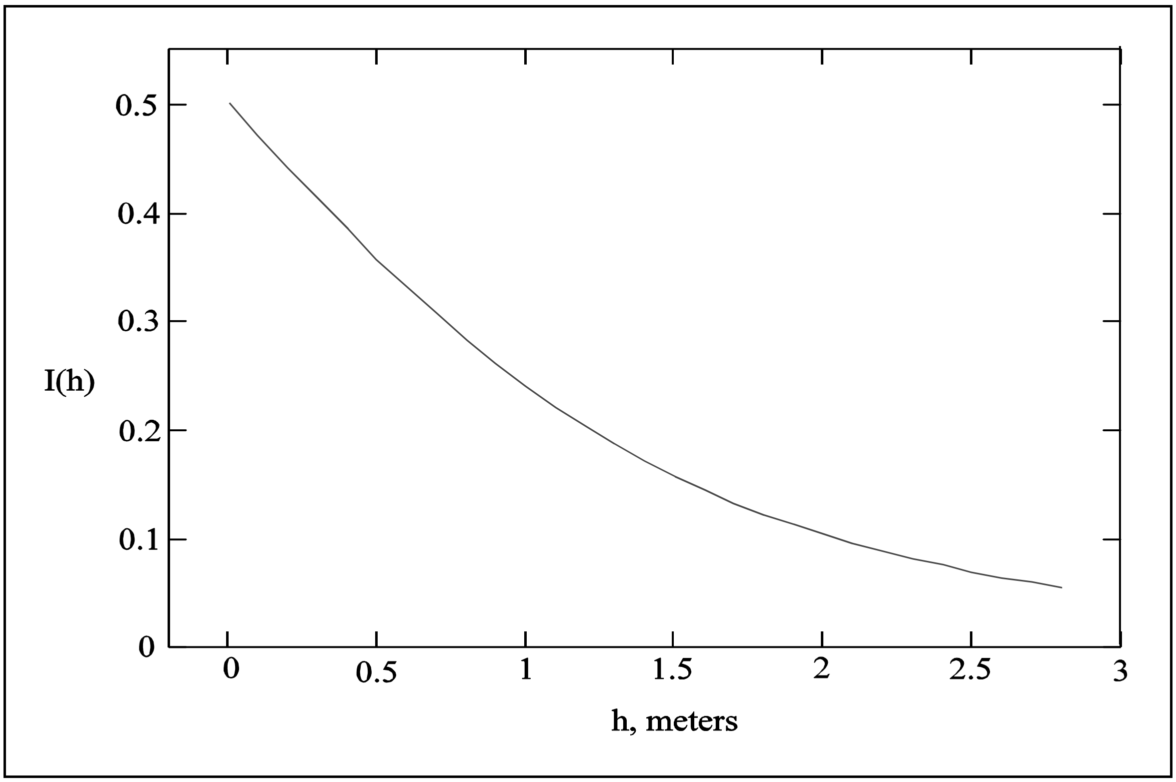

Figure 3. Numerical evaluation of I(h).

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[3] The reference to a commercial product is for technical communication only, and does not constitute an endorsement of the product.

[4] If the instrument is much closer to the shield, which would make the calculated factor larger, it should be possible to make the shield smaller, which would lower the heat input. These two effects should approximately cancel.

III. Extension to a swept back shield.

It may prove desirable to sweep back the shield by an angle α, as shown in Fig. 4. Such a change in geometry may be required to adjust the torque on the spacecraft from solar radiation pressure. A non-zero α would have two effects on the thermal input to the patch on the cylinder wall. First, it would change F(h). Second, it would cause the shield to have a position-dependent temperature, with the pattern tied to the Sun's azimuth. We separately consider these below.

A seen from a fixed point on the cylinder, the swept back shield would cover a larger portion of the sky than one that was flat. The sky blockage by the shield depends only on the distance $h' = h - u$, where *u* is the distance (in the -X direction) by which the outer edge of the shield has been shifted toward the dark end of the cylinder. (Depending on how the variation of α was implemented, r_2 might be a function of α. This will not be addressed explicitly below.)

$$F_{p-s}(h) = I(h-u) \tag{5}$$

But, for parts of the cylinder closer to the Sun-facing end than *u*, this will not work. For this case, the calculation of Section II can be taken to yield F for the sky, and (for $h-u < 0$)

$$F_{p-s}(h) = 1 - I(u-h) \tag{6}$$

The sweep back will change the shield temperature. The primary effect is the change in the angle ζ between the shield normal and the direction to the Sun. This results in a shield temperature that depends on ν, where ν is the Sun's azimuth with respect to the point on the shield. In particular, the power input to the shield is proportional to $\cos(\zeta)$:

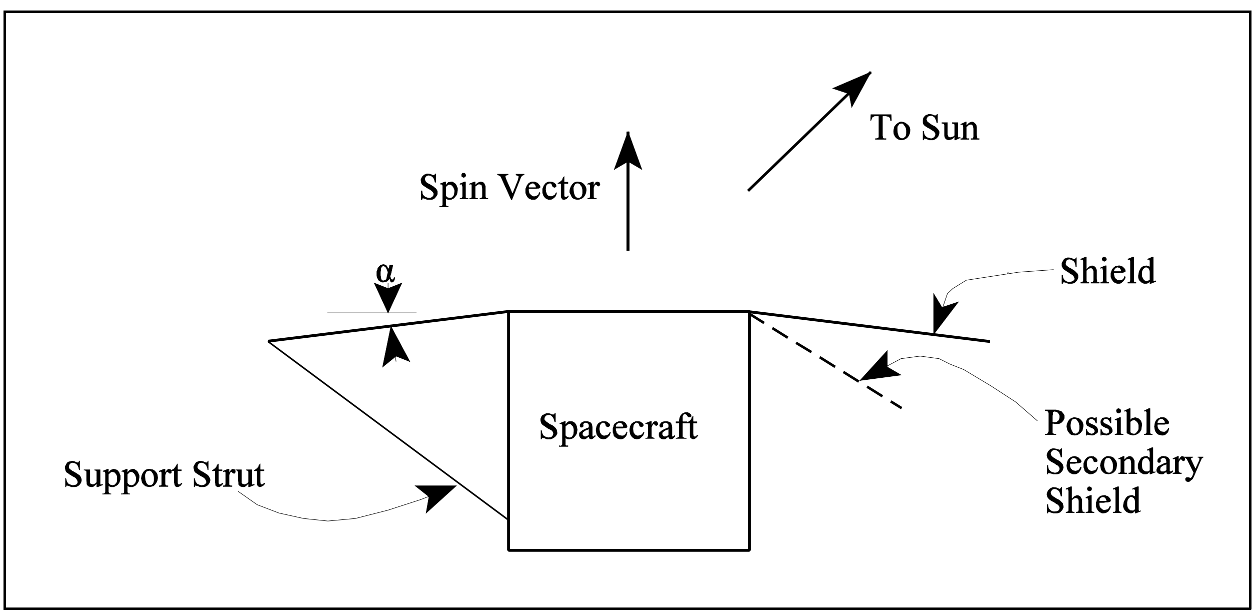

Figure 4. Spacecraft with shield swept back by α. An auxiliary shield is also shown.

$$\cos(\zeta) = \cos(\alpha)\cos(\xi) + \sin(\alpha)\sin(\xi)\cos(\nu) \qquad \textbf{(7)}$$

Of the two terms on the right hand side, the first yields an uninterestingly small shift in temperature for reasonable values of α (say, $|\alpha| < 10$ deg). The second results in a time variation of the shield temperature with spacecraft rotation. For $\alpha = 2$ deg,[5] the 8 W/m^2 received from the shield at the instrument has a periodic component of about 0.25 W/m^2 amplitude that could cause a temperature variation of 0.05 deg. (It will be seen in Section V that this is small compared to the effect of Earth radiation.)

The second effect that changes the shield temperature is that the back of the shield will see less sky, and will therefore not cool as well. (The sky seen will be replaced by spacecraft, which will be reflective and thus mostly like sky.) This second effect will not be considered here since it is small (for the scale of α likely to be used) and it primarily affects the overall shield temperature and not its variation with ν.

IV. Use of an auxiliary shield.

Based on the above discussion, it may be desirable to decrease the flow of heat through the shield to the instrument. This might be the case if the heat from the shield had a significant time-dependent component. One approach would be to replace (or back) the single-layer shield with a blanket of MLI. This would provide a several-fold reduction in the heat flow.

An alternative approach would be to introduce a secondary shield as shown in Fig. 4. The sweep back angle, α', would need to be optimized. Making it large increases the view of space by the facing surfaces of the two shields, making the shielding effect more efficient. However a large α' increases $F_{p\text{-}ss}$, the GCF for heat transfer from the second shield to a patch on the cylinder wall, i.e., increases the heat transfer to the instrument.

The effect of the secondary shield on the thermal input from Earth would need to be considered. (See also, next section.) To minimize heating by Earth when its direction is far from the spin axis (i.e., Earth is near the observation plane), α' would be made large, limited by the requirement that the telescope field of view not be obstructed. The required optimization of α' is not addressed here. However, this geometry would increase the heating by Earth when it is near the anti-Sun direction. The latter could be reduced by imposing a small scale “staircase shape” on the anti-Sun side of the secondary shield. This approach requires further investigation, particularly as it would affect heat radiated by the spacecraft.

An alternative configuration for the secondary shield would again be a layer of material

[5] It was shown in TM97-03 that the torque due to radiation pressure (on the shield) goes through a null at $\alpha \approx 1.7$ deg for a nominal set of parameters. This sets the scale of interesting values of α. See also Reasenberg and Phillips, SPIE 3356-38, in press 1998.

extending from the cylinder wall to the ring represented by the outer edge of the shield depicted in Fig. 4. The distinguishing feature is that the alternative shield would meet the spacecraft along a circle shifted toward the dark end of the spacecraft. This approach would yield a region on the cylinder wall where varying amounts of excess heat could be radiated without impacting the temperature-sensitive instrument at the anti-Sun end.

A possible advantage to a secondary heat shield is the immunity it offers to leakage paths past the primary heat shield. Such leakage could come from small holes (manufacturing defects, small impacts, etc.), gaps between sections of the shield, and reflections between overlapping sections of heat shield. A quantitative assessment will be needed.

V. Thermal radiation from Earth.

Here we assume the Earth either reflects of reradiates the heat it receives from the Sun. Then the average emission is S/4 = 340 W/m^2. Taking this emission to be isotropic is clearly an overly simplified description. The power absorbed from Earth may be increased by a factor of about 2 when the spacecraft is on the Sun-lit side and there is a difference in both the reflected sunlight and the thermal IR received by the spacecraft between being over the pole and being over the sub-solar point. For a spacecraft at 16 R_e from the Earth (15 R_e above the surface, where R_e is the radius of Earth), the GCF is easily shown to be F = 0.004 for a nadir-pointing surface. The corresponding heat input on a black surface facing Earth is 1.36 W/m^2. Note that the heat input from Earth will vary at the spacecraft rotation rate and will be modulated by both the spacecraft orientation with respect to the nadir and the location of the sub-Earth point. To radiate an extra 1.36 W/m^2, a black surface at 300 K must increase temperature by 0.2 K.

This variable heat input is unavoidable for a spinning spacecraft in Earth orbit. Insulation wrapped around the spacecraft will mitigate the effect of this input, except for the radiation that enters one of the two the ports required for the starlight to reach the optical system. The fluctuation of the heat received from Earth is large compared to the expected fluctuation in the heat received from the Sun shield, which suggests at first that no further improvement is needed in the latter. However, there is likely to be a higher correlation with parts of the sky observed with the Sun azimuth than with the Earth azimuth. Thus, it will be desirable for the time-dependent thermal input from the Sun to be made smaller than that from the Earth.

VI. Discussion

A thin solar shield is shown to substantially reduce the thermal input and (more importantly) the temporal variation of the thermal input to the instrument from the Sun. The unshielded instrument would receive directly from the Sun from 0 to 960 W/m^2 as the spacecraft rotated. With the flat shield, the solar input is constant at 24 W/m^2 at the shield end and 5 W/m^2 at the anti-shield end. With the shield swept back by 2 deg, the solar input at 1.5 m from the shield end ranges from 7.75 to 8.25 W/m^2 over the period of rotation (20 min). Further, for a 10 day precession period (assuming that the precession is driven by radiation pressure), there is a comparable variation over the period of precession. The combined fluctuation range is three orders lower than without the shield. However, the Earth can contribute from 0 to ≈ 3 W/m^2 to the cylinder surface, and to its dark end. Thus, with the nominal design, the largest rapidly (i.e., at the spin rate) fluctuating source of external heat is Earth.

Finally, we note that all the fluctuations in input heat are at the level of a few watts, well within the control authority of any expected thermal-control system. Thus, suppressing the astrometric effects of these thermal fluctuations will depend on the effective gain of active thermal-control systems, the passive suppression associated with thermal time constants, and the immunity of the astrometric system to thermal variations. The passive suppression is enhanced by the increase in the spacecraft spin rate, while the active suppression is likely limited by leakage paths rather than by loop gain.

Should the Earth thermal load be an unacceptable problem, consideration could be given to an L2 orbit, 0.01 AU from Earth, and on the anti-Sun side. The Earth would be on the sun-side of the solar shield, and thus not a problem. The largest remaining source of heat would be the Moon, which would provide about three orders less thermal burden than Earth at 100,000 km. This 15 fold increase in distance from Earth would have an impact on telecommunications, but Earth would be visible all of the time from a phased array mounted flat on the Sun-shield end of the cylinder.

VII. Acknowledgments

I thank J.D. Phillips for reviewing this memorandum in draft form.

VIII. Distribution

FAME web site *via* S. Horner

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